IECEC '95-28

EPS PERFORMANCE PREDICTIONS SUPPORT TOPEX/POSEIDON SPACECRAFT MISSION BEYOND DESIGN GOAL OF 5 YEARS

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ABSTRACT

TOPEX/Poseidon is a scientific satellite, launched successfully on August 10, 1992 to map the surface topology of the earth's oceans. The electrical power system for this satellite consists of the following elements: a single wing, solar array, a solar array drive & power transfer assembly, a peak power tracker. and storage batteries. The satellite's electronics arc designed for a tlmcc-year primary mission. Because of a potential mission extension, the solar array, batteries, and propellant arc sized for a five year mission goal. This paper addresses an overview, of the EPS and then performance predictions beyond five years arc made. The EPS will meet all the mission requirements of the TOPEX/Poseidon Spacecraft for seven years.

INTRODUCTION

The TOPllX/Poseidon Satellite, herein abbreviated TOPEX (ocean Topography Experiment), measures the earth's ocean surface topography (wave heights) from space using radar altimeters. TOPEX was launched from the Kourou Space Center in french Guiana into a nominal circular orbit with an altitude of 1334 Km and an inclination of 66 degrees. The satellite electronics are designed for a three year primary mission. Because of a potential mission extension, the solar anay (S/A). batteries, and propellant arc sized for a five year mission.

Organization. TOPEX/Poseidon is a joint mission between NASA and the French CNES, in support of the World Climate Research Program. JPL manages the project for the NASA Office of Space Science and Applications, As the TOPEX prime contractor under JPL, Fairchild Space (now Orbital Sciences Corporation) Designed, built, integrated, and tested the satellite. The power subsystem engineering and the solar array subsystem design were performed by Fairchild.

Electrical Power Subsystem

A diagram of the TOPEX satellite is shown in Figure-l. A block diagram of the Electrical Power Subsystem (EPS) is shown in Figure-2. S/A power is transferred through the S/A drive assembly via sliprings. The standard power regulator unit (SPRU) within the MPS serves as the power processing interface between the S/A and the satellite load. Three 50 AH batteries located in the MI'S supply power whenever the load requirements exceed the SPRU output, and during SLIn occult aliens,

EPS PERFORMANCE PREDICTIONS

Solar Array

Tile solar array is designed to provide about 1043 watts of power to the satellite loads after processing through the MI'S, at the end of five years in orbit and its flight performance is excellent.

There are three main reasons why the solar array is expected to support TOPEX mission (while supplying the spacecraft full load power) beyond five years.

Reason-1. During the TOPEX PDR, the direction/ recommendation was to employ a radiation design margin (RDM) of iwo anti later on it dropped to 1.6. This is equivalent to a RDM of 2 for the first three years of the mission and 1 for the fourth and fifth years. This recommendation might be interpreted by some to result in a conservative design, but by

This work was performed for the Jet Propulsion Laboratory, California Institute of Technology, Sponsored by the National Aeronautics and Space Administration,

others as a cautious design approach. This is because there was no previous satellite which flew in this orbit; this RDM might in fact be useful in prolonging the mission if the electronics last beyond five years. Also note that the TOPEX/Poseidon employed Class-S part program. Thus, it can be interpreted that there was "experience and fore-lhoogh(" behind the direction 10 use an RDM of 1.6. Solar array flight performance [1] indicates there is no additional degradation and hence, the usc of the RDM of 1.6 will result in about 5% more power from the solar array at the end of life of five years,

Reason-2. During the PDR and the CDR design phases, it was difficult 10 constrict the launch date since unexpected events may occur and the launch can be delayed. Hence, the S/A has to be designed assuming an unconstrained launch date. This resulted in the assumption of a summer solstice launch. Thus, the TOPEX/Poseidon launch on August 1 (), 1992 resulted in about 1% more power at the end of life of five years.

Reason-3. The spacecraft power consumption at the end of five years is computed to bc 1043 waits and at the beginning of life 10 be about 1000 walls. However, current spacecraft orbital average power consumption seems to bc around 850 watts. As the spacecraft ages, varioos degradations take place, i.e., heater power consumption increases, efficiency of various electronic units degrade resulting in more power demand, charge efficiency and end of night voltage (Figure-3) of the batteries degrade with cycle life, solar panels ron al high temperature at EOL compared 10 BOL, etc. Thus power demand is expected to increase 10 about 970 walls and this results in about 7% more power at the end of life of five years.

Performance Analysis. As part of the EPS design and evaluation, Fairchild developed the computer program "POWERT" [2]. This program enables onc to compute/predict the solar array performance. under various operating conditions and mission life durations. This program was originally validated using the ground measured data and recently telemetered/ measured solar array output data [1]. Thus, the "POWERT" computer program is used 10 predict the solar array performance at the end of 6, 7, and 8 years. Various solar array degradation factors assumed in (his analysis arc tabulated in Table- 1. As the solar array degrades with life, lower and lower amounts of incident solar energy is converted into electricity, and more energy is dissipated on the solar array thereby resulting in higher solar panel operating temperatures. As the battery end of orbital night voltage dccrcascs/degrades with cllargc-discharge cycle life, SPRU in-take energy during each sunrise decreases.

Results. All these factors are included in predicting the solar array performance, and the results of these analyses arc presented in columns 1 to 5 of Table-2. The last column of this table presents tbc estimated spacecraft power demand. It is very clear from this table that the spacecraft power demand can be met for more than 7 years assuming all other electronics remain intact.

Storage Batteries

Fairchild's extensive in-house study resulted in selecting the MMS bus for meeting the TOPEX/Poseidon mission requirements while maintaining flight heritage. The Modular Power Subsystem (MPS) part of the MMS bus has a provision to house up to three NASA Standard 50 AH Nickel-~acfnliLln~ (22 cell) batteries. This design did not have the provision for cell bypass in case of cell degradation. As no single point failure should jeopardize the mission, energy storage batteries were sized to completely take care of a single point failure. Two 22-cell 50 AH Nickel Cadmium batteries arc adequate to meet the energy storage requirements while satisfying the depth of discharge and end of night battery bus voltage at the end of five years [3]. However, 10 satisfy the single point failure requirement, three such batteries were employed,

Baseplate imbedded heatpipes force all three batteries to operate at the same temperature, thereby eliminating the degradation otherwise induced by the temperature differentials between the three batteries. All three batteries arc performing in an excellent manner.

Due to higher than normal differential half battery voltages observed on-board UARS, GRO, and other satellites, implementation of certain operational measures were thought to be very appropriate even though the battery related parameters and circumstances were not identical. in view of this, the following operational changes were made: (i) limited peak charge current 1020 amps maximum; (ii) limited overcharge by controlling the recharge fraction, namely, charge/discharge (C/D) ratio to 1.05 +/- 0.03 at 6°C; (iii) limited taper charge current during full sunlight periods 10 less than 200 mA; and (iv) use LOW current sensor data rather than HIGH current sensor data 10 improve the C/D ratio computational accuracy when the battery currents arc equal or lower than 3 amps. These measures might have further assured excellent performance being exhibited by the batteries.

Ground Testing. Usually a certain number of cells (from the same 10 I as the flight cells) arc set aside for ground testing. The main purpose of the ground tests is to obtain an advanced warning of any anomaly or degradation. This advance information allows precautionary measures to be taken with respect to flight battery operational modes, environment, etc to avoid or minimize soch anrm~aly/degradation.

To date the TOPEX test cells, under-going ground tests (1 .ifc/Stress Test, Mission Simulation Test, and Temperature effect Test)[4], have exhibited results comparing favorably with results from traditionally "good" cells.

Analysis. The TOPEX/Poseidon cells were manufactured according to the NASA Standard 50 Ampere-Hour Cell Specification with additional Mandatory Inspection Points. The ground Cell tests and on-orbit performance of the TOPEX/Poseidon batteries reaffirmed that the TOPEX/Poseidon cells in fact fall under the category of traditional "good" cells. These facts further substantiated the use of the NASA Battery Life Predictions (DOD Vs Cycle Life) as presented in Figure-4. In addition to cycle life, the cod of night (discharge) voltage is essential for operation of the spacecraft. Table-3 summarizes the data from the life testing conducted (at NSWC, Crane) on NASA

Standard 50 AI I cells. Figure-3 presents the end of night (discharge) voltage Vs cycle life extrapolated using the ground test data from TOPEX/Poseidon cells.

Case-A: No Cell/Battery Degradation/Failure. First the anticipated power demand from the TOPEX/Poseidon is estimated and battery depth of discharge for each case is computed. Next, the battery life corresponding to each DOD is read from the Figure-4. The results are summarized in Table-4. From this table, it can be concluded that the energy storage require.mcnk can be met for up 10 8 years.

Case-B: After a Single Point Failure (One battery is disconnected from the bus). A single point failure can occur at any time. Thus, current analysis is conducted assuming that a single point failure can occur al end of 2 years 10 8 years. The anticipated power demand from the TOPEX/Poseidon is estimated first. Next, the battery depth of discharge is computed for two conditions; (a) before the occurrence of a single point failure, and (b) after the occurrence of a single point failure. Now, the battery life is computed using the data from Figure-4.

From the summarized results, in Table-5, it can be concluded that the energy storage requirements can be met for 6 10 7 years depending upon when a failure occurs.

Power Electronics

Proper design reliability has been incorporated in addition to enhancing parts reliability by proper derating, stress level reduction, and employing Class-S parts. Adequate judicious redundancy has also been incorporated in the design of the peak power tracker and other electronics.

All power electronics are performing in superb manner. Measured peak power tracking accuracy is better than predicted and convulsion efficiency is higher than predicted.

CONCLUSIONS

Thus, from the above predictions it is obvious that the EPS can meet all the mission requirements of the TOPEX/Poseidon Spacecraft for al least seven years.

ACKNOWLEDGEMENTS

The work described in this paper was performed by Fairchild Space Company under contract Number 957849 with the Jet propulsion Laboratory, Pasadena, California.

The authors gratefully acknowledge the contributions of the many individuals al the Jet Propulsion Laboratory anti Fairchild Space. Special thanks to Alfred R. Zieger, the TOPEX/Poseidon Project Mission Manager at JPL anti Rex Richardson, the TOPEX/Poseidon Program Manager at Fairchild.

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1 “

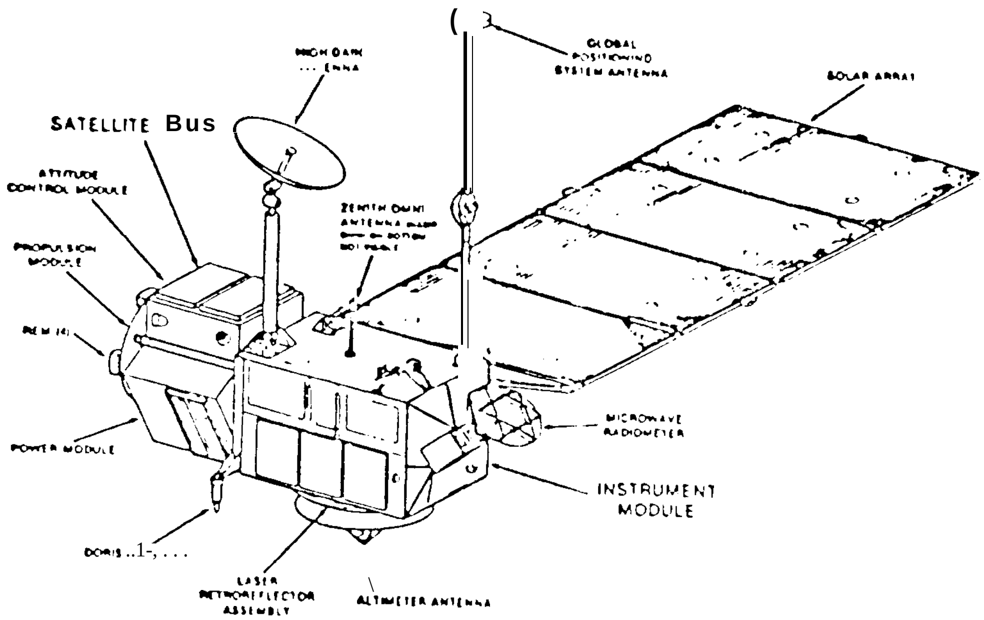


FIGURE-1

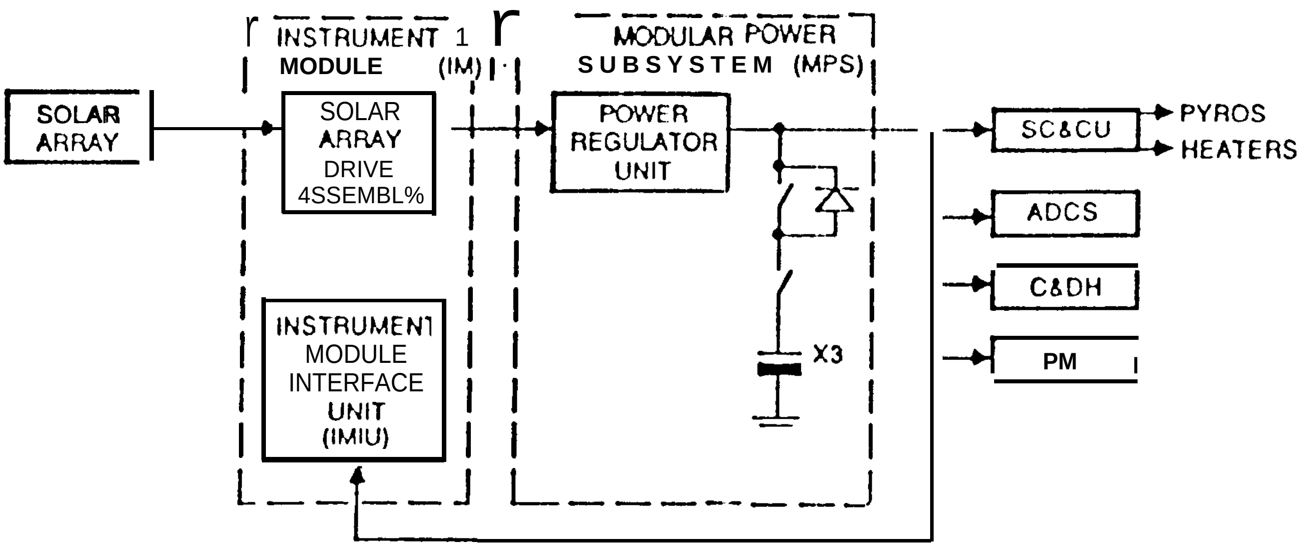


FIGURE-2

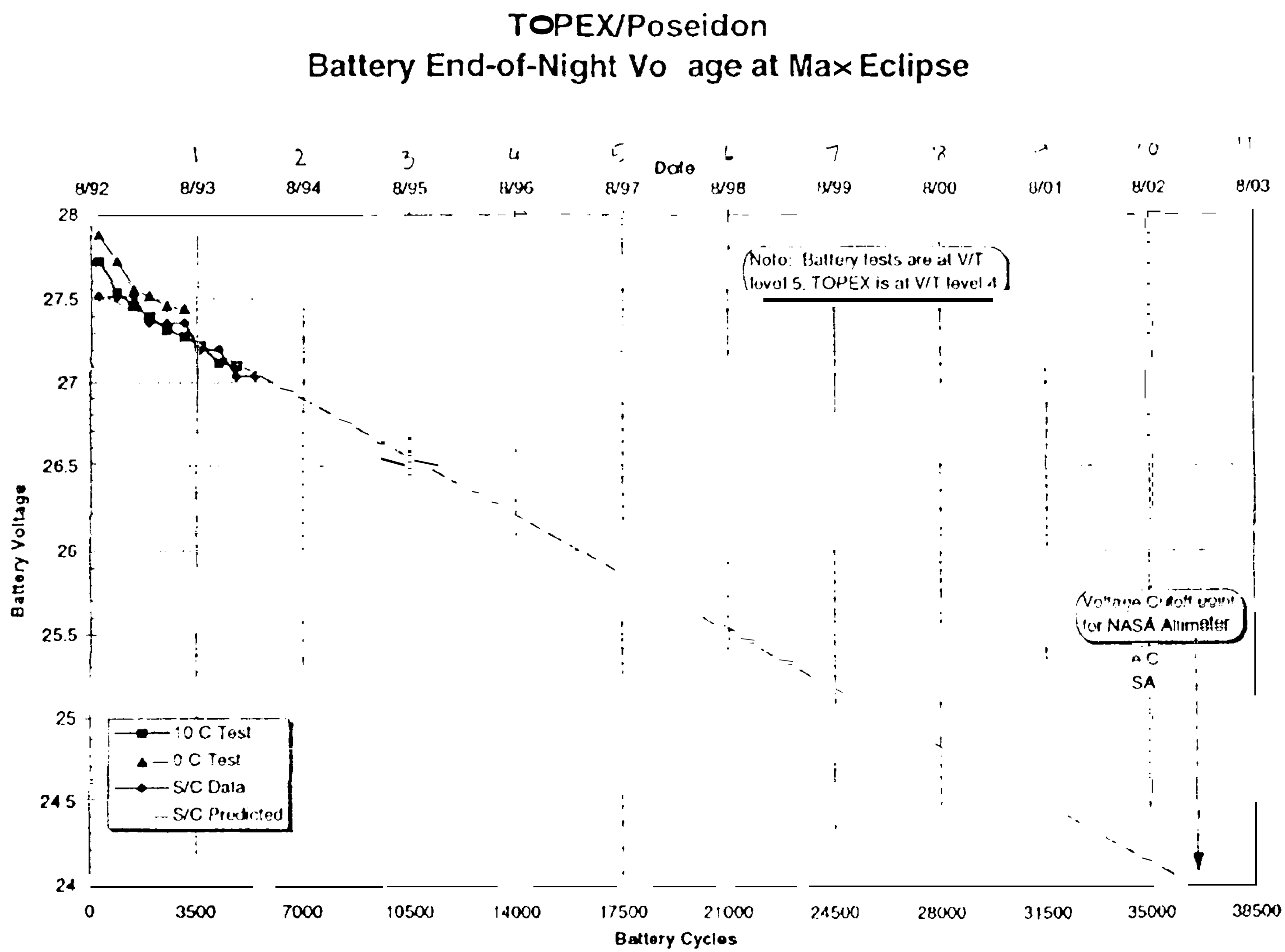
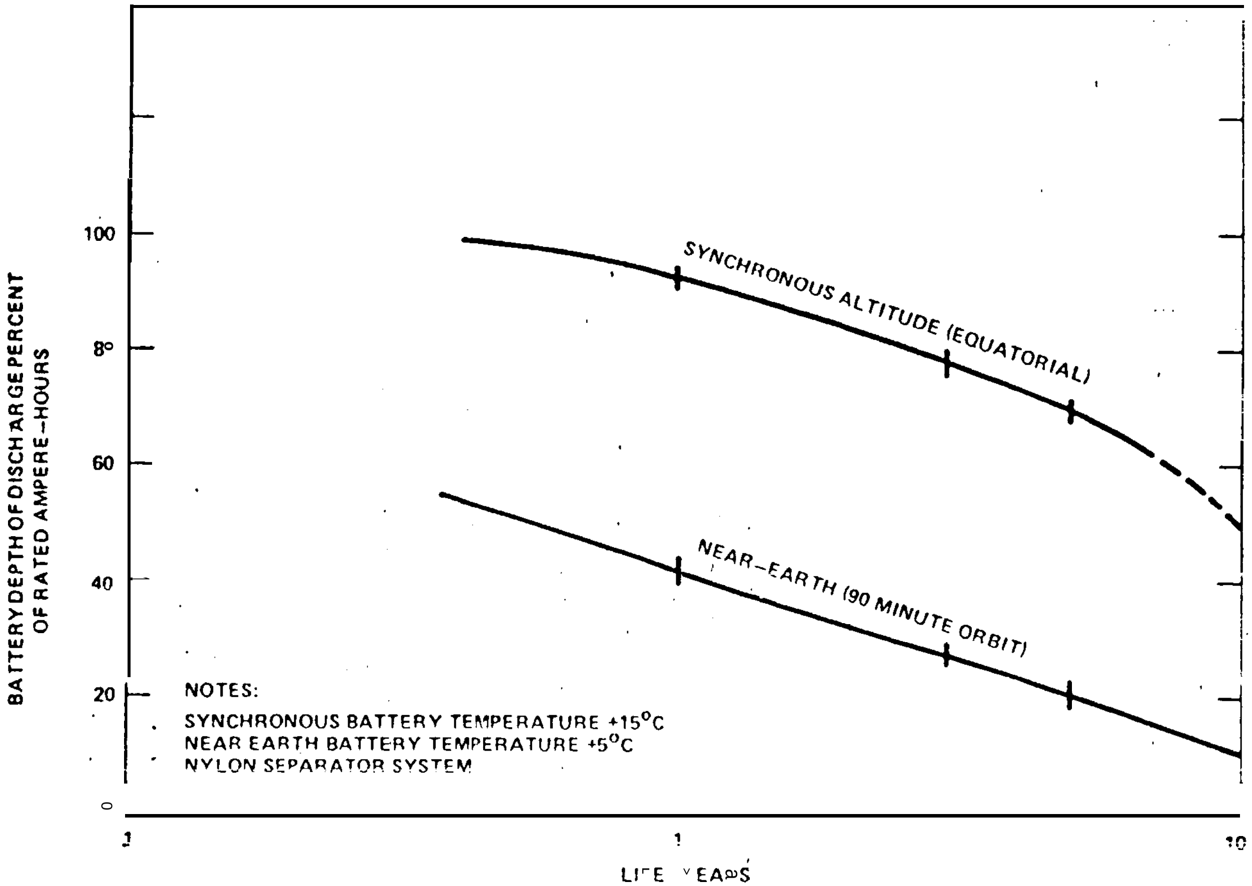

TOPEX/Poseidon
Battery End-of-Night Vo age at Max Eclipse
Date
8/92
8/93
8/94
8/95
8/96
8/97
8/98
8/99
8/00
8/01
8/02
8/03
Battery Voltage
Battery Cycles
Note: Battery tests are at V/T level 5. TOPEX is at V/T level 4
for NASA Altimeter
SA
10 C Test
0 C Test
S/C Data
S/C Predicted

FIGURE-3



MAXIMUM DESIGN UTILIZATION OF NICKEL-CADMIUM BATTERIES FOR SPACECRAFT APPLICATIONS

FIGURE-4

Table-1

Solar Array Degradation Factors					
Mission Duration (Yrs)	UV	MM	Thermal Cycling	Radiation	Cumulative Degradation
0.5	0.9925	0.9990	0.9980	0.9100	0.9005
1.0	0.9900	0.9980	09960	0 8700	08561
1.5	0.9888	0.9970	0.9940	08450	0.8280
2.0	0.9875	0.9960	09920	0 8260	08059
3.0	0.9850	0.9940	0.9880	0 8000	07739
4.0	09825	0.9920	09840	0 7820	07.500
5.0	0.9800	D, Woo	09800	..07450	0.7274
6.0	0.9800	0.9880	0.9760	0 7520	0 7106
7.0	0.9800	0.9860	09720	0 7400	06950
8 0	09800	09840	0.9680	c|	7300 0.6814 —

Table-2

Solar Array Life Prediction to supply load to TOPEX ocecraft					
1	2	3	4	5	6
Conditions/Life	Battery Nite Voltage (V)	S/A output (at MPS inpu (W)	P (available to load) (W)	P(load req) (Design) (W)	P(consumec (W)
Topex Design :Sym, SS sum, SS temp 2.3% d... RDM=1.6	25.8	2139	1043	1043	930 0s of 4 /94
EOL= 5 Yrs, ss sun, ss temp (radiation, RDM= 1)		2231	10$8	1043	
EOL= 5 Yrs, Aug 10, ss temp (radiation, RDM= 1)		2252	1099	1043	970 (estimated)
EOL= 6 Yrs, Aug 10, ss femp Above plus 2.5 C hotter Above plus lower current limi (radiation, RDM= 1)	**255**	2164	1056	1043	995 (estimated)
EOL= 7 Yrs, Aug 10, ss temp Above plus 4.6 C hotter Above plus lower current limit (radiation, RDM= 1)	**25.2**	2069	**1009**	1043	1010 (estimated)
EOL= 8 Yrs., Aug 10, ss temp Above plus 2.5 C heifer = WS Above plus lower current limit (radiation, RDM= 1)	**24.8**	196$'	960	1043	102U (estimated)

Table-3

NASA Standard 50 AH Ni-Cd Cell Evaluation Data	
Performance Parameters	Crane Life Test Data
Cycle Period	1.48 Hrs
Temperature	20C
Depth of Discharge	25%
Total Cycles Completed	27533
Cell Discharge Voltage	1.138 volts

Table- 4

/ Mission Duration (Y	ption: No cell (Power Demand (Estimated)(W)	battery degra Depth of Dischagre (%)	dation or foi Battery Life (Yrs)	e End of Night Voltage
5	970	13.5	>8	Acceptable
6	W5	13.8	>8	Acceptable
7	1010	14.0	>8	May be
8	10'20	14,2	>8	May be

Table- 5

ssumption: Single Point Failure - One battery Off Spacecraft Power Bus

Mission Duration (Yrs)	Power Demand (Estimated)(W)
2	930
3	945
4	955
5	970
6	W5
7	1010
8	1020

SPF Occurred at EOL (Yrs)	DOD-Before SPF (%)	DOD After SPF (%)	Battery Life (Yrs)	End of Night Voltage
2	12.92	19.38	6.1	Acceptable
3	13.13	19.69	6.3	Acceptobk
4	13.26	19.90	6.6	Acceptable
5	13.47	20.21	6.9	Accepf able
6	13.82	2073	7.1	Moy be
7	14.03	21.04	7.6	May be
8	14.17	21.25	>8.0	May be